TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 625

SPINNING CHARACTERISTICS OF WINGS

IV - CHANGES IN STAGGER OF RECTANGULAR CLARK Y

BIPLANE CELLULES

By M. J. Bamber and R. O. House
Langley Memorial Aeronautical Laboratory

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SUMMARY

Rectangular Clark Y biplane cellules having zero and
-0.25 stagger, the gap equal to the chord, and 0° decalage
were tested on the N.A.C.A. spinning balance in the 5-foot
vertical tunnel. The aerodynamic characteristics of the
models and a prediction of the angles of sideslip for
steady spins for airplanes using these wing arrangements
are given. There is included an estimation of the yawing
moment that must be furnished by parts of the airplane
other than the wing to balance the inertia couples and wing
yawing moments for spinning equilibrium. The effects on
the spin of changes in stagger and of variation in some of
the important parameters are discussed and the results are
compared with those for a similar biplane cellule with 0.25
positive stagger tested earlier.

It is concluded that, with the values of stagger con-
sidered, for a conventional biplane having equal upper and
lower rectangular Clark Y wings, gap equal to the chord,
and zero decalage: The airplane will generally spin with
inward sideslip, which, in some cases, may exceed 20°; for
angles of attack through 50°, the sideslip generally will
become more inward as the stagger becomes more negative
and, for an angle of attack of 70° and sometimes of 60°,
the inward sideslip will become less as the stagger becomes
more negative; the value of stagger for the best spinning
characteristics will vary with different types of airplanes;
the provision of a yawing-moment coefficient of -0.025 (i.e.,
opposing the spin) by the tail, fuselage, and interference
effects will prevent equilibrium in a steady spin for the
values of stagger tested and with any of the parameters
used in the analysis; and too much reliance should not be
placed on tail arrangement for preventing bad spinning
characteristics.

INTRODUCTION

In order to provide necessary aerodynamic data for estimating airplane spinning characteristics from the design features, the N.A.C.A. is conducting an investigation to determine the aerodynamic characteristics of airplane models and parts of airplane models in spinning attitudes.

The portion of the investigation to determine the spinning characteristics of wings, for which the N.A.C.A. spinning balance is being used, includes a study of the effects of variations in airfoil section, plan form, and tip shape of monoplane wings and variations in stagger for biplane cellules. The first and third series of tests reported were made of Clark Y monoplane wings with rectangular plan forms, with square and rounded tips, and with a 5:2 tapered wing having rounded tips (references 1 and 2). The second series, made of a rectangular Clark Y biplane cellule with 0.25 stagger, is reported in reference 3.

The present report is a continuation of reference 3 and gives the aerodynamic characteristics, in spinning attitudes, of a rectangular Clark Y biplane cellule with the gap equal to the chord, zero decalage, and with zero and -0.25 stagger. Also included are comparisons with the cellule having 0.25 positive stagger.

APPARATUS AND MODELS

The tests were made on the spinning balance in the N.A.C.A. 5-foot vertical wind tunnel. The tunnel is described in reference 4 and the six-component spinning balance in reference 5.

The Clark Y wings were made of laminated mahogany with balsa insets for lightness. The span of each wing is 30 inches and the chord is 5 inches. These wings had been used for the tests in reference 3; the only change in the cellule was new strut bracing to give the desired amounts of stagger. Figure 1 is a sketch of the model showing the bracing, balance attachment, and stagger. Figure 2 shows the model (-0.25 stagger) mounted on the spinning balance.

TESTS

In order to cover the probable spinning range, tests were made at angles of attack of 30°, 40°, 50°, 60°, and 70°. At each angle of attack tests were made with values of $\Omega b/2V$ of 0.25, 0.50, 0.75, and 1.00. At each angle of attack and at each value of $\Omega b/2V$ tests were made at sideslip angles of -5°, 0°, 5°, 10°, and 15° for the cellule with zero stagger, and at 0° ($\alpha = 70^{\circ}$ only), 5°, 10°, 15°, and 20° sideslip for the cellule with -0.25 stagger. The angles of attack and of sideslip were measured in the plane of symmetry at the quarter-chord point of the upper wing, which was also the center of rotation for all tests. Because of variations in individual balance readings, at least one repeat test was made for each condition and an average of the individual measurements was used to compute the coefficients.

The tunnel air speed was 70 feet per second for tests with $\frac{\Omega b}{2V} = 0.25$ and 0.50, 56 for $\frac{\Omega b}{2V} = 0.75$, and 42 for $\frac{\Omega b}{2V} = 1.00$. The Reynolds Number was about 180,000 for the highest air speed and about 137,000 for the lowest. Previous tests showed no appreciable change in scale effects for this range.

RESULTS AND DISCUSSION

The data were converted to coefficient form by means of the following relations:

$$C_X = \frac{X}{qS} \qquad C_Y = \frac{Y}{qS} \qquad C_Z = \frac{Z}{qS}$$

$$C_l = \frac{L}{qbS} \qquad C_m = \frac{M}{qbS} \qquad C_n = \frac{N}{qbS}$$

All coefficients are standard N.A.C.A. form except C_m, which is based on the span rather than on the chord and may be converted to the standard N.A.C.A. form by multiplying by 6. All coefficients are given with the conventional signs for right spins (reference 1).

The coefficients of longitudinal force in the earth system of axes $C_{X''}$ and of all six components of the forces and moments in the body system of axes are given in tables I and II. Sample curves of $C_{X''}$, C_l, C_m, and C_n are given in figures 3 to 6.

The data and attitudes are given for the quarter-chord point on the lower surface of the upper wing at zero radius. The coefficients in body axes may be converted to any other point of rotation in the plane of symmetry by the following relations. The converted coefficients are indicated by the subscript 1.

$$C_{X_1} = C_X \left(\frac{V}{V_1}\right)^2 \qquad C_{Y_1} = C_Y \left(\frac{V}{V_1}\right)^2 \qquad C_{Z_1} = C_Z \left(\frac{V}{V_1}\right)^2$$

$$C_{l_1} = \left[C_l + \frac{z}{b} C_Y\right] \left(\frac{V}{V_1}\right)^2$$

$$C_{m_1} = \left[C_m - \frac{z}{b} C_X + \frac{x}{b} C_Z\right] \left(\frac{V}{V_1}\right)^2$$

and

$$C_{n_1} = \left[C_n - \frac{x}{b} C_Y\right] \left(\frac{V}{V_1}\right)^2$$

where x is the distance forward (positive) of the new center of rotation from the quarter-chord of the upper wing.

   z, the distance of the new center of rotation below (positive) the lower surface of the upper wing.

   b, the span of the wing.

$$\frac{V_1}{V} = \sqrt{\frac{u_1^2}{V^2} + \frac{v_1^2}{V^2} + \frac{w_1^2}{V^2}}$$

$$\frac{u_1}{V} = \cos \alpha \cos \beta + \frac{2zq}{b\Omega} \left(\frac{\Omega b}{2V}\right)$$

$$\frac{v_1}{V} = \sin \beta + \frac{2xr}{b\Omega} \left(\frac{\Omega b}{2V}\right) - \frac{2zp}{b\Omega} \left(\frac{\Omega b}{2V}\right)$$

$$\frac{w_1}{V} = \sin \alpha \cos \beta - \frac{2xq}{b\Omega} \left(\frac{\Omega b}{2V}\right)$$

$$\frac{p}{\Omega} = \cos \alpha \cos \beta$$

$$\frac{q}{\Omega} = \sin \beta$$

$$\frac{r}{\Omega} = \cos \beta \sin \alpha$$

Thus $\quad \alpha_1 = \tan^{-1} \dfrac{w_1}{u_1}$

$$\beta_1 = \sin^{-1} \frac{v_1}{V_1}$$

An analysis was made with the data converted to the quarter-chord point midway between the wings of the biplane (reference 3). The analysis showed that the sideslip required was generally about 2° less than it was for the original data. In other details the variations were quite similar.

The data are believed to be correct to within the following limits:

$C_{X''}$, ±0.02; $\quad C_X$, ±0.02; $\quad C_Y$, ±0.01; $\quad C_Z$, ±0.02;

C_l, ±0.001; C_m, ±0.002; C_n, ±0.001

No corrections have been made for the effects of jet boundaries, scale, or interference of the balance, struts, or bracing system.

Generally, $C_{X''}$ decreases as the stagger decreases (fig. 3). This result may normally be expected because of the blanketing of the upper wing by the lower wing. The variation of C_l with stagger changes sign with increase of angle of attack (fig. 4). The values of C_l at 30° angle of attack are more positive for the negative stagger, and at 70° angle of attack are more positive for the positive stagger. The changes in C_l with $\Omega b/2V$ are irregular. The values of C_m increase as the stagger decreases (fig. 5). Part of this increase is due to measuring the moments about a fixed point on the upper wing so that de-

creasing the stagger means moving the lower wing forward
with respect to this fixed point. (See fig. 1.) The val-
ues of C_n are small and show no general tendency to
change with the stagger (fig. 6).

ANALYSIS

An analysis of the data was made to show the effects
of certain parameters on the steady spinning characteris-
tics of an airplane using these types of biplane cellule.
The method of analysis with the assumptions used and the
errors involved is given in reference 1. For convenience
the method of analysis is briefly described.

Formulas used in the analysis.—

$$\frac{\Omega b}{2V} = \sqrt{\frac{-C_m}{3.84\,\mu\,\sin 2\alpha} \times \frac{b^2}{k_Z^2 - k_X^2}} \qquad (1)$$

$$C_l = C_L \left(\frac{k_Z^2 - k_Y^2}{b\sqrt{k_Z^2 - k_X^2}} \right) \sqrt{\frac{-C_m\,\tan\,\alpha}{2\mu}}$$

$$+ 1.02 \left(\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} \right) \left(\frac{-C_m\,\sin\,\beta}{\cos\,\alpha} \right) \qquad (2)$$

$$C_n = C_l\,\cot\,\alpha \left(\frac{k_Y^2 - k_X^2}{k_Z^2 - k_Y^2} \right) \qquad (3)$$

Parameters.— Because the wing loading, aspect ratio,
radii of gyration, and pitching moments are mostly depend-
ent upon the characteristics of the particular airplane,
values of these variables covering the range for normal
biplanes have been used in the analysis. A mean of these
values was chosen that gave the following parameters:

Relative density of the airplane to air

$$\left(\mu = \frac{W}{g\rho bS} = \frac{M}{\rho bS} \right) \mu = 5$$

Pitching-moment inertia parameter, $\dfrac{b^2}{k_Z^2 - k_X^2} = 80$

Rolling-moment and yawing-moment inertia parameter,

$$\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$$

Slope of assumed pitching-moment curve for the complete airplane, $\dfrac{-C_m}{\alpha - 20^o} = 0.0020$

Lift coefficient $C_L = C_{X''}$ from test data

Each of the parameters was varied, one at a time, from the mean value while all others were kept at the mean value with the exception of C_L, which was equal to $C_{X''}$ for all cases. The values of the parameters chosen are:

$\mu = 2.5$, 5.0, 7.5, and 10.0

$\dfrac{b^2}{k_Z^2 - k_X^2} = 60$, 80, 100, and 120

$\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 0.5$, 1.0, 1.5, and 2.0

$\dfrac{-C_m}{\alpha - 20^o} = 0.0010$, 0.0015, 0.0020, 0.0025, and 0.0030

The variations in μ include the range of wing loadings of conventional biplanes.

The variations in $\dfrac{b^2}{k_Z^2 - k_X^2}$ and $\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ cover the range given in reference 6 for 11 airplanes. These parameters may be written $\dfrac{W b^2}{g(C - A)}$ and $\dfrac{C - B}{C - A}$, respectively, where

$A = mk_X{}^2$,   the moment of inertia about the   X   axis

$B = mk_Y{}^2$,   the moment of inertia about the   Y   axis.

$C = mk_Z{}^2$,   the moment of inertia about the   Z   axis.

Method of analysis.- The value of $\Omega b/2V$ was computed
for each angle of attack using equation (1). The aerody-
namic rolling-moment coefficient required for spinning
equilibrium was computed for all values of α and β
tested using equation (2). The values of C_m and μ were
those used in equation (1). In order to obtain values of
C_L ($C_L = C_{X''}$), values of $C_{X''}$, determined from the tests,
were plotted against $\Omega b/2V$ and, by interpolation, values
of $C_{X''}$ at the values of $\Omega b/2V$ computed from equation
(1) were found. By means of similar interpolation, values
of C_l were obtained; a correction of 0.02 was added to
C_l to give C_l available. The values of C_l available
and of C_l required, as found by the preceding methods,
were plotted against β, the points of intersection of
the two sets of curves giving values of C_l and β, for
each angle of attack, at which all forces and moments ex-
cept yawing moments are in equilibrium.

Values of C_n required to balance the inertia yawing
moments were calculated from equation (3), using for C_l
the value found for the equilibrium condition. The value
of C_n furnished by the wings was the C_n of the tests
corrected by adding 0.006. By the subtraction of this
value of C_n from the value of C_n required as found by
equation (3), the value of C_n was found that must be
furnished by the remaining parts of the airplane, fuse-
lage, empennage, and interference effects, to give equi-
librium in a steady spin at the given angles of attack.

Scale-effect corrections to C_l ($\Delta C_l = 0.02$) and to
C_n ($\Delta C_n = 0.006$) have been found necessary from compari-
sons of model with full-scale data and are discussed in
reference 5.

Discussion of results of analysis.- The angles of
sideslip required for a balance of rolling moments and the
values of C_n that must be supplied by parts of the air-

plane to balance the inertia couples and wing yawing mo-
ments are plotted against the parameters in figures 7 to
14. The results for the 0.25 stagger are included for
comparison.

The effect of the various parameters on the sideslip
required for equilibrium of rolling moments depends on the
angle of attack and on the amount of stagger (figs. 7 to
10). For 50° angle of attack and below, the sideslip is
always inward and, except for two cases, is never less
than 6°, generally increasing as the stagger decreases
(changes in a direction from positive to negative). For
an angle of attack of 70° and, sometimes, of 60°, β de-
creases as the stagger decreases, and for some parameters
the sideslip may become outward.

The effect of stagger on C_n required is small. (See
figs. 11 to 14.) The C_n required tends to change in the
direction from positive toward negative as the stagger in-
creases. The variation of C_n required with the parame-
ters $\dfrac{-C_m}{\alpha - 20°}$, μ, and $\dfrac{b^2}{k_Z^2 - k_X^2}$ is usually small, the

maximum negative value of C_n required being less than

-0.016. The C_n required decreases as $\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ in-

creases, the extreme values being 0.013 and -0.023.

Prediction of spinning characteristics of an airplane
from the analysis.- Prediction of the spinning characteris-
tics of an airplane in which any of these biplane combina-
tions is used largely depends upon the aerodynamic yawing-
moment characteristics of the particular airplane. The
value of C_n required, as given in this report, is numer-
ically equal and of opposite sign to the sum of the wing
yawing moments and the inertia couples. At any angle of
attack, when this value of C_n is supplied by the empen-
nage, fuselage, and interference effects, a steady spin
will result provided that the equilibrium is stable; for
any other value of C_n the airplane will not spin at that
attitude. In order to insure against spinning in any atti-
tude, a value of C_n opposing the spin must be provided
that is larger than any attainable value of C_n required.
The yawing moment supplied by the empennage, fuselage, and

interference effects depends upon the sideslip; the size
and shape of the fuselage and tail surfaces; the location
of the horizontal tail surfaces with respect to the fuse-
lage, fin, and rudder; the amount of fin area ahead of
the center of gravity; the interference effects between
the wings and the rest of the airplane; and the limits of
control movements. Data on some of these effects are re-
ported in reference 5 and in references 7 to 12. The ge-
ometry of the spin indicates that the vertical tail sur-
faces should become more effective in producing a yawing
moment opposing the spin as the rate of rotation increases
and the sideslip decreases. Fin area ahead of the center
of gravity will give yawing moments opposing the spin if
the sideslip is inward. (See reference 11, fig. 2.)

If the effects of sideslip on the yawing moment sup-
plied by the fuselage, empennage, and interference effects
are not considered, for values of stagger tested a biplane
with negative stagger will generally have a slightly small-
er yawing moment than one with positive stagger. When

$\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} < 1$ (weight of the airplane distributed along

the fuselage, $A < B$), the C_n required opposing the spin

will be smallest. When $\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} > 1$ (weight of the air-

plane distributed along the wings, $A > B$), the C_n re-
quired opposing the spin will be large and the airplane
may be expected to spin flat and recoveries will probably
be more difficult.

The effects of sideslip on the yawing moments pro-
duced by the tail and fin area indicate that, with large
inward sideslip, the vertical tail surfaces would be very
ineffective and large amounts of fin area ahead of the
wing would be beneficial. In some cases the inward side-
slip at the center of gravity may be large enough to make
the sideslip at the tail inward, in which case the tail
and the fuselage behind the center of gravity would furnish
yawing moments aiding the spin. It follows that two gen-
eral methods of preventing a dangerous spin might be con-
sidered.

The first method is to design an airplane that will
attain spinning equilibrium with as small an amount of in-
ward sideslip as possible so that the rear part of the

fuselage and the tail surfaces will have maximum effective-
ness. A tail with a large unshielded vertical fin area
will then give the maximum obtainable yawing moment oppos-
ing the spin. A large diving moment, a small value of
$\dfrac{b^2}{k_Z{}^2 - k_X{}^2}$, a large value of wing loading, and a large
value of $\dfrac{k_Z{}^2 - k_Y{}^2}{k_Z{}^2 - k_X{}^2}$ are factors giving the smallest
amounts of inward sideslip, although the large values of
$\dfrac{k_Z{}^2 - k_Y{}^2}{k_Z{}^2 - k_X{}^2}$ also give relatively large values of C_n re-
quired opposing the spin.

The second method is based on the assumption that an
appreciable yawing moment opposing the spin may be set up
by fin area ahead of the center of gravity (reference 11).
This yawing moment would be expected to increase as the
inward sideslip and the vertical fin area ahead of the
center of gravity increase. The airplane should then be
designed with the maximum possible vertical fin area ahead
of the center of gravity; and, to obtain maximum inward
sideslip, a small diving moment, a large value of
$\dfrac{b^2}{k_Z{}^2 - k_X{}^2}$, lightly loaded wings, and a small value of

$$\dfrac{k_Z{}^2 - k_Y{}^2}{k_Z{}^2 - k_X{}^2}.$$

A good tail arrangement, i.e., one with a large un-
shielded fin area, may not always prevent flat spins be-
cause, for some cases, the sideslip at the tail may be
zero or inward, which will result in a tail yawing moment
of zero or even aiding the spin.

CONCLUSIONS

On the assumption that the arbitrary constants added
to the rolling-moment and yawing-moment coefficients are

of the right order of magnitude, the following conclusions are indicated by the analysis presented for a conventional biplane with rectangular Clark Y wings having 0.25, zero, and -0.25 stagger, gap equal to the chord, and 0° decalage:

1. The value of the yawing-moment coefficient required from the fuselage, tail, and interference effects for steady spinning equilibrium at any angle of attack is small and nearly always negative (opposing the rotation) throughout the range investigated.

2. The maximum value of the yawing-moment coefficient that must be supplied by all parts of the airplane other than the wings and inertia couples to prevent spinning equilibrium at any angle of attack is $C_n = -0.025$.

3. The value of stagger for the best spinning characteristics varies with different types of airplanes.

4. At some angles of attack, the inward sideslip will be very great (more than 20°) so that even good tail arrangements may have little effect in preventing a dangerous spin; fin area ahead of the wings will be beneficial.

5. The angle of attack at which the maximum inward sideslip occurs decreases as the stagger changes from positive toward negative. For angles of attack through 50°, the sideslip generally becomes more inward as the stagger becomes more negative, the opposite being true at 70° angle of attack, with the transition taking place at some intermediate angle of attack.

6. Too much reliance should not be placed on tail arrangement for preventing bad spinning characteristics.

Langley Memorial Aeronautical Laboratory,
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TABLE I. Aerodynamic Characteristics of a Clark Y Biplane Cellule
— 0.25 stagger

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n	$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
				$\beta = 5^\circ$									$\beta = 15^\circ$				
0.25	30	0.777	-0.028	-0.005	-0.904	0.0008	0.0028	0.0038	0.25	30	0.737	-0.050	-0.001	-0.890	-0.0342	0.0006	0.0089
	40	.688	-.025	-.009	-.918	.0046	.0055	-.0019		40	.655	-.036	.000	-.887	-.0333	.0035	-.0018
	50	.518	.000	-.008	-.802	.0252	.0095	-.0043		50	.500	-.024	.001	-.807	-.0159	.0085	-.0030
	60	.397	.042	-.003	-.722	.0136	.0093	-.0046		60	.318	-.036	.001	-.677	-.0061	.0092	-.0037
	70	.299	.062	-.001	-.675	-.0061	.0088	-.0013		70	.300	-.020	.005	-.643	-.0230	.0049	-.0010
.50	30	.828	-.025	.005	-.964	.0083	.0063	.0054	.50	30	.778	-.033	.000	-.917	-.0117	.0046	.0043
	40	.633	-.057	-.009	-.874	.0870	.0057	-.0025		40	.650	-.015	-.009	-.874	.0041	.0053	-.0008
	50	.529	-.039	-.011	-.857	.0331	.0062	-.0067		50	.503	-.023	-.009	-.810	.0333	.0069	-.0065
	60	.430	.000	-.009	-.840	.0298	.0077	-.0059		60	.402	.001	-.002	-.801	.0282	.0090	-.0064
	70	.390	.028	-.001	-.789	.0146	.0087	-.0031		70	.295	.037	.000	-.754	.0056	.0053	-.0032
.75	30	.884	-.031	.017	-1.039	-.0404	.0038	.0006	.75	30	.840	-.034	.013	-.989	-.0271	-.0051	.0007
	40	.800	-.053	.003	-1.088	.0235	.0037	.0013		40	.750	-.018	.003	-.995	.0142	-.0054	-.0007
	50	.621	-.045	-.005	-1.020	.0178	.0025	-.0039		50	.647	.013	-.002	-.991	.0134	-.0062	-.0041
	60	.509	-.020	-.010	-1.053	.0272	.0000	-.0056		60	.531	.033	-.007	-1.004	.0242	-.0033	-.0047
	70	.365	.008	-.001	-1.044	.0132	-.0021	-.0030		70	.388	.043	-.003	-1.018	.0183	-.0025	-.0009
1.00	30	.986	-.019	.032	-1.150	-.1290	-.0006	-.0048	1.00	30	1.030	.001	.031	-1.189	-.1029	-.0175	-.0014
	40	.974	-.049	.024	-1.313	-.0313	-.0055	.0016		40	1.012	.018	.024	-1.306	-.0254	-.0166	.0009
	50	.819	-.044	.018	-1.325	-.0118	-.0036	.0007		50	.845	.037	.009	-1.270	-.0089	-.0161	-.0009
	60	.637	-.034	.003	-1.333	-.0103	-.0117	-.0083		60	.666	.049	.010	-1.285	.0014	-.0179	.0000
	70	.534	.023	-.004	-1.488	-.0281	-.0331	.0008		70	.558	.077	.001	-1.430	.0068	-.0157	.0006
				$\beta = 10^\circ$									$\beta = 20^\circ$				
0.25	30	0.757	-0.031	-0.005	-0.893	-.0164	0.0023	0.0052	0.25	30	0.792	-0.024	0.000	-0.928	-0.0465	-0.0008	0.0052
	40	.655	-.036	-.005	-.886	-.0161	.0051	-.0014		40	.697	-.011	.005	-.919	-.0476	.0023	-.0010
	50	.508	-.012	-.001	-.801	.0053	.0108	-.0032		50	.453	-.029	.005	-.785	-.0339	-.0059	.0017
	60	.369	.021	.002	-.703	.0040	.0100	-.0050		60	.397	.037	.004	-.730	-.0352	.0074	-.0028
	70	.284	.057	.003	-.672	-.0165	.0054	-.0010		70	.299	.063	.008	-.700	-.0389	.0037	-.0010
.50	30	.774	-.038	.003	-.916	-.0009	.0057	.0050	.50	30	.739	-.045	.003	-.880	-.0262	-.0002	.0044
	40	.628	-.043	-.008	-.855	.0164	.0055	-.0030		40	.627	-.036	-.006	-.849	-.0121	.0008	.0007
	50	.490	-.045	-.012	-.816	.0369	.0062	-.0065		50	.487	-.035	-.008	-.799	.0156	.0044	-.0048
	60	.402	-.009	-.001	-.819	.0288	.0079	-.0059		60	.391	-.004	-.003	-.790	.0285	.0065	-.0051
	70	.284	.034	.001	-.768	.0104	.0065	-.0030		70	.281	.006	.000	-.744	-.0035	.0041	-.0023
.75	30	.870	-.027	.013	-1.020	-.0337	.0030	.0001	.75	30	.779	-.051	.008	-.930	-.0304	-.0018	.0002
	40	.772	-.046	-.001	-1.046	.0196	.0041	-.0009		40	.701	-.037	.000	-.938	.0069	.0046	-.0035
	50	.624	-.035	-.008	-1.013	.0151	.0014	-.0049		50	.581	-.018	-.003	-.923	.0145	-.0012	-.0038
	60	.489	-.022	-.013	-1.016	.0208	.0040	-.0060		60	.497	.013	-.007	-.972	.0315	.0043	-.0062
	70	.331	-.015	-.005	-1.008	.0140	.0008	-.0022		70	.350	.011	-.004	-.994	.0304	.0024	-.0021
1.00	30	.989	-.021	.039	-1.153	-.1132	-.0027	-.0033	1.00	30	.880	-.057	.023	-1.049	-.0893	-.0114	-.0016
	40	.987	-.029	.023	-1.313	-.0326	-.0034	-.0005		40	.862	-.034	.015	-1.163	-.0248	-.0013	-.0030
	50	.802	-.033	.006	-1.287	-.0104	-.0051	-.0010		50	.775	.015	.011	-1.188	-.0093	-.0118	-.0003
	60	.618	-.033	-.006	-1.293	-.0054	-.0068	-.0014		60	.619	.013	.000	-1.341	.0057	-.0036	-.0051
	70	.471	-.018	-.004	-1.420	-.0116	-.0177	-.0001		70	.467	.004	.003	-1.354	.0160	-.0073	-.0013

Coefficients of forces and moments given for and about the quarter-chord point at the lower surface of the upper wing.

$\frac{\Omega b}{2V}$	α (deg.) R	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
					$\beta = 0^\circ$			
0.25	70	0.280	0.053	-0.002	-0.877	0.0095	0.0048	-0.0013
.50	70	.297	.033	-.001	-.807	.0128	.0045	-.0030
.75	70	.364	-.002	-.008	-1.070	.0154	-.0145	-.0028
1.00	70	.833	.028	.010	-1.499	-.0368	-.0455	.0010

TABLE II. Aerodynamic Characteristics of a Clark Y Biplane Cellule
Zero stagger

Ωb/2V	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n	Ωb/2V	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
				$\beta = 0^\circ$									$\beta = 10^\circ$				
0.25	30	0.833	-0.042	-0.007	-0.988	0.0158	-0.0228	0.0061	0.25	30	0.809	-0.041	-0.005	-0.958	-0.0154	-0.0241	0.0052
	40	.753	-.040	-.012	-1.017	.0135	-.0270	-.0008		40	.737	-.035	-.010	-.992	-.0233	-.0374	-.0015
	50	.611	-.030	-.009	-.988	.0205	-.0248	-.0014		50	.637	-.013	.001	-1.006	-.0145	-.0270	-.0013
	60	.438	-.010	-.003	-.889	.0357	-.0248	-.0017		60	.448	-.009	.004	-.913	.0064	-.0244	-.0011
	70	.394	.031	-.003	-.774	.0312	-.0258	-.0035		70	.374	.018	.003	-.766	.0124	-.0255	-.0028
.50	30	.917	-.033	.007	-1.078	-.0060	-.0280	.0054	.50	30	.874	-.040	.002	-1.032	-.0181	-.0282	.0045
	40	.813	-.054	-.006	-1.090	.0222	-.0315	.0033		40	.774	-.026	-.008	-1.032	-.0009	-.0388	.0037
	50	.629	-.043	-.016	-1.030	.0493	-.0305	-.0011		50	.616	-.016	-.011	-.980	.0196	-.0283	-.0021
	60	.449	-.054	-.011	-.958	.0507	-.0300	-.0045		60	.480	.001	-.005	-.959	.0371	-.0331	-.0036
	70	.304	-.009	-.004	-.815	.0473	-.0325	-.0042		70	.304	-.003	.000	-.897	.0496	-.0311	-.0042
.75	30	1.017	-.051	.023	-1.193	-.0639	-.0300	.0026	.75	30	.998	-.039	.015	-1.175	-.0582	-.0389	.0010
	40	.951	-.033	.011	-1.299	.0177	-.0390	.0044		40	.905	-.034	.004	-1.210	.0050	-.0405	.0020
	50	.776	-.017	-.005	-1.227	.0468	-.0453	.0011		50	.713	-.018	-.008	-1.130	.0276	-.0440	.0006
	60	.584	-.037	-.013	-1.174	.0499	-.0458	-.0027		60	.552	-.006	-.013	-1.135	.0390	-.0480	-.0024
	70	.380	-.014	-.013	-1.150	.0559	-.0450	-.0037		70	.380	-.003	-.007	-1.120	.0497	-.0446	-.0033
1.00	30	1.111	-.027	.043	-1.299	-.1576	-.0383	-.0006	1.00	30	1.141	-.039	.033	-1.334	-.1339	-.0425	.0000
	40	1.148	-.001	.036	-1.497	-.0541	-.0530	.0064		40	1.096	-.040	.025	-1.464	-.0364	-.0630	.0036
	50	.917	-.030	.023	-1.461	.0109	-.0678	.0087		50	.905	-.009	-.007	-1.419	.0051	-.0668	.0042
	60	.641	-.067	.002	-1.398	.0257	-.0603	.0019		60	.707	.009	-.005	-1.398	.0232	-.0619	.0019
	70	.475	-.025	-.007	-1.455	.0397	-.0650	-.0002		70	.475	-.011	-.006	-1.413	.0505	-.0665	.0030
				$\beta = 5^\circ$									$\beta = 15^\circ$				
0.25	30	0.823	-0.038	-0.008	-0.973	0.0013	-0.0237	0.0055	0.25	30	0.818	-0.037	-0.005	-0.964	-0.0322	-0.0244	0.0050
	40	.746	-.039	-.012	-1.007	.0036	-.0272	-.0013		40	.746	-.029	-.002	-.998	-.0388	-.0379	-.0010
	50	.623	-.033	-.007	-.995	.0040	-.0253	-.0017		50	.637	-.021	.007	-1.016	-.0355	-.0274	-.0015
	60	.448	-.001	.000	-.898	.0225	-.0237	-.0014		60	.457	-.009	.004	-.939	-.0122	-.0254	-.0006
	70	.365	.005	.000	-.751	.0216	-.0255	-.0031		70	.351	-.009	.001	-.760	-.0002	-.0253	-.0022
.50	30	.895	-.037	.002	-1.055	-.0131	-.0259	.0047	.50	30	.842	-.050	.001	-1.001	-.0255	-.0284	.0043
	40	.792	-.035	-.008	-1.063	.0101	-.0301	.0023		40	.747	-.053	-.008	-1.002	-.0157	-.0396	.0019
	50	.634	-.029	-.015	-1.008	.0339	-.0296	-.0017		50	.610	-.033	-.007	-.975	.0068	-.0309	-.0014
	60	.463	-.019	-.011	-.960	.0436	-.0308	-.0040		60	.450	-.006	-.001	-.929	.0315	-.0318	-.0038
	70	.296	-.016	-.003	-.910	.0459	-.0310	-.0039		70	.306	.006	-.003	-.878	.0407	-.0319	-.0038
.75	30	1.028	-.029	.031	-1.201	-.0593	-.0343	.0019	.75	30	.939	-.053	.015	-1.115	-.0565	-.0334	.0010
	40	.978	-.011	.006	-1.283	.0145	-.0418	.0031		40	.866	-.030	.000	-1.156	-.0031	-.0390	.0015
	50	.750	-.010	-.008	-1.179	.0382	-.0489	.0006		50	.708	.002	-.007	-1.098	.0209	-.0393	.0003
	60	.607	.018	-.012	-1.184	.0470	-.0436	-.0023		60	.536	-.008	-.011	-1.086	.0343	-.0390	-.0038
	70	.374	-.014	-.012	-1.131	.0519	-.0484	-.0038		70	.394	.006	-.005	-1.106	.0544	-.0409	-.0033
1.00	30	1.154	-.016	.038	-1.341	-.1439	-.0441	.0001	1.00	30	1.110	-.054	.030	-1.313	-.1204	-.0459	.0008
	40	1.158	.007	.029	-1.505	-.0439	-.0589	.0044		40	1.084	-.007	.019	-1.431	-.0303	-.0551	.0017
	50	.944	.001	.016	-1.469	.0045	-.0638	.0037		50	.866	-.001	.008	-1.348	.0029	-.0595	.0030
	60	.764	-.043	.004	-1.432	.0239	-.0619	.0015		60	.740	.052	-.007	-1.390	.0237	-.0597	.0009
	70	.462	-.026	-.012	-1.490	.0453	-.0594	.0003		70	.481	.006	-.009	-1.391	.0516	-.0562	-.0001

Coefficients of forces and moments given for and about the quarter-chord point at the lower surface of the upper wing.

(side table) $\beta = -6^\circ$

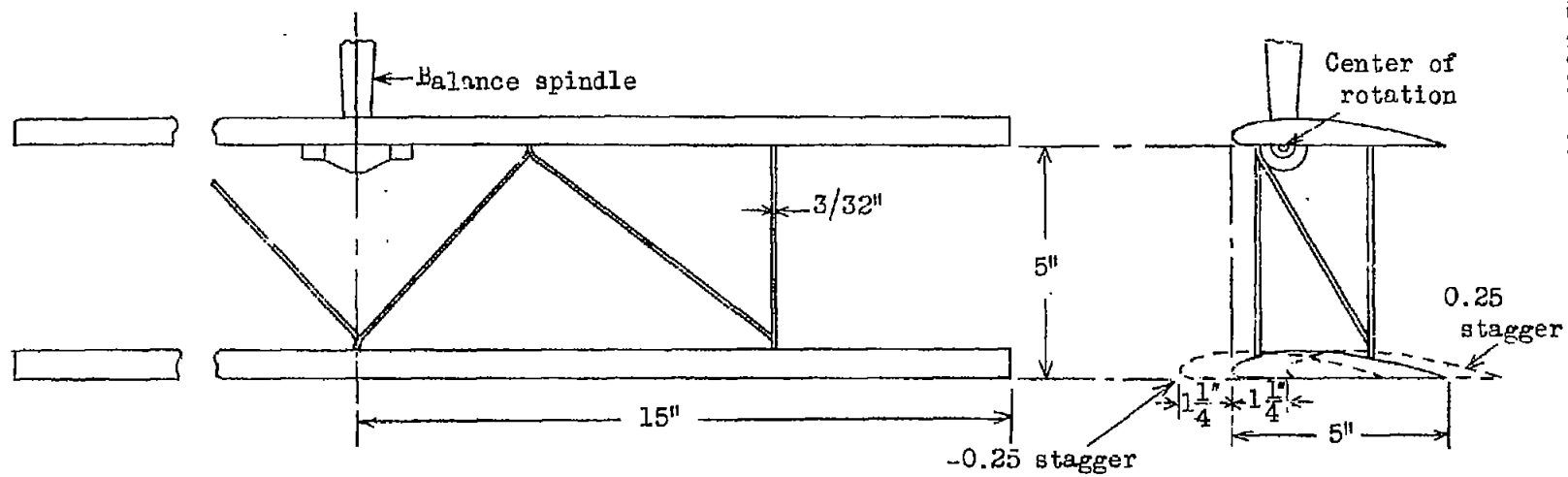| Ωb/2V	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
0.25	30	0.875	-0.057	-0.008	-1.044	0.0300	-0.0838	0.0058
	40	.806	-.016	.034	-1.065	.0282	-.0737	.0006
	50	.659	.010	-.011	-1.029	.0345	-.0645	.0036
	60	.475	.024	-.008	-.910	.0547	-.0381	.0022
	70	.502	.018	-.007	-.810	.0518	-.0314	.0017
.50	30	.972	-.058	.015	-1.145	-.0018	-.0497	.0054
	40	.864	-.053	.004	-1.171	.0331	-.0530	.0035
	50	.881	-.035	-.008	-1.088	.0531	-.0474	-.0008
	60	.477	-.023	-.015	-.980	.0635	-.0435	-.0035
	70	.316	-.003	-.007	-.831	.0560	-.0380	-.0038
.75	30	1.046	-.008	.026	-1.298	-.0060	-.0613	.0019
	40	.980	.002	.015	-1.209	.0223	-.0558	.0060
	50	.818	.015	-.004	-1.268	.0389	-.0531	.0019
	60	.610	.004	-.015	-1.283	.0613	-.0651	-.0025
	70	.416	-.007	-.008	-1.236	.0558	-.0688	-.0035
1.00	30	1.207	-.034	.059	-1.515	-.1327	-.0384	-.0034
	40	1.204	-.039	.044	-1.471	-.0387	-.0387	-.0009
	50	.888	-.031	.034	-1.535	.0109	-.0730	.0038
	60	.783	-.031	.002	-1.449	.0351	-.0700	.0008
	70	.554	.039	-.003	-1.408	.0314	-.0603	.0008

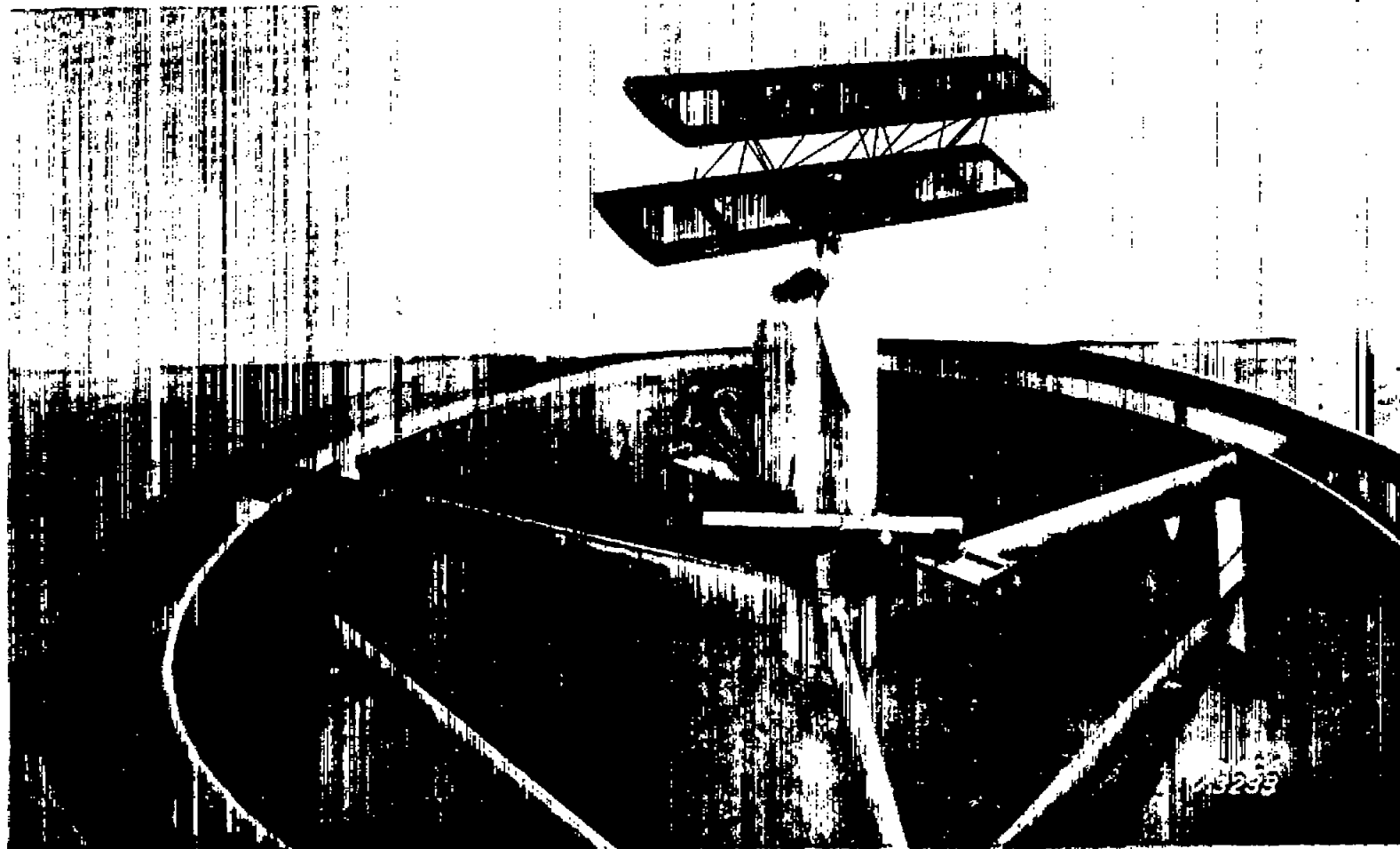Figure 1.- Clark Y biplane cellule.

Fig. 1

Figure 2.- The rectangular Clark Y biplane cellule, 0.25 stagger,
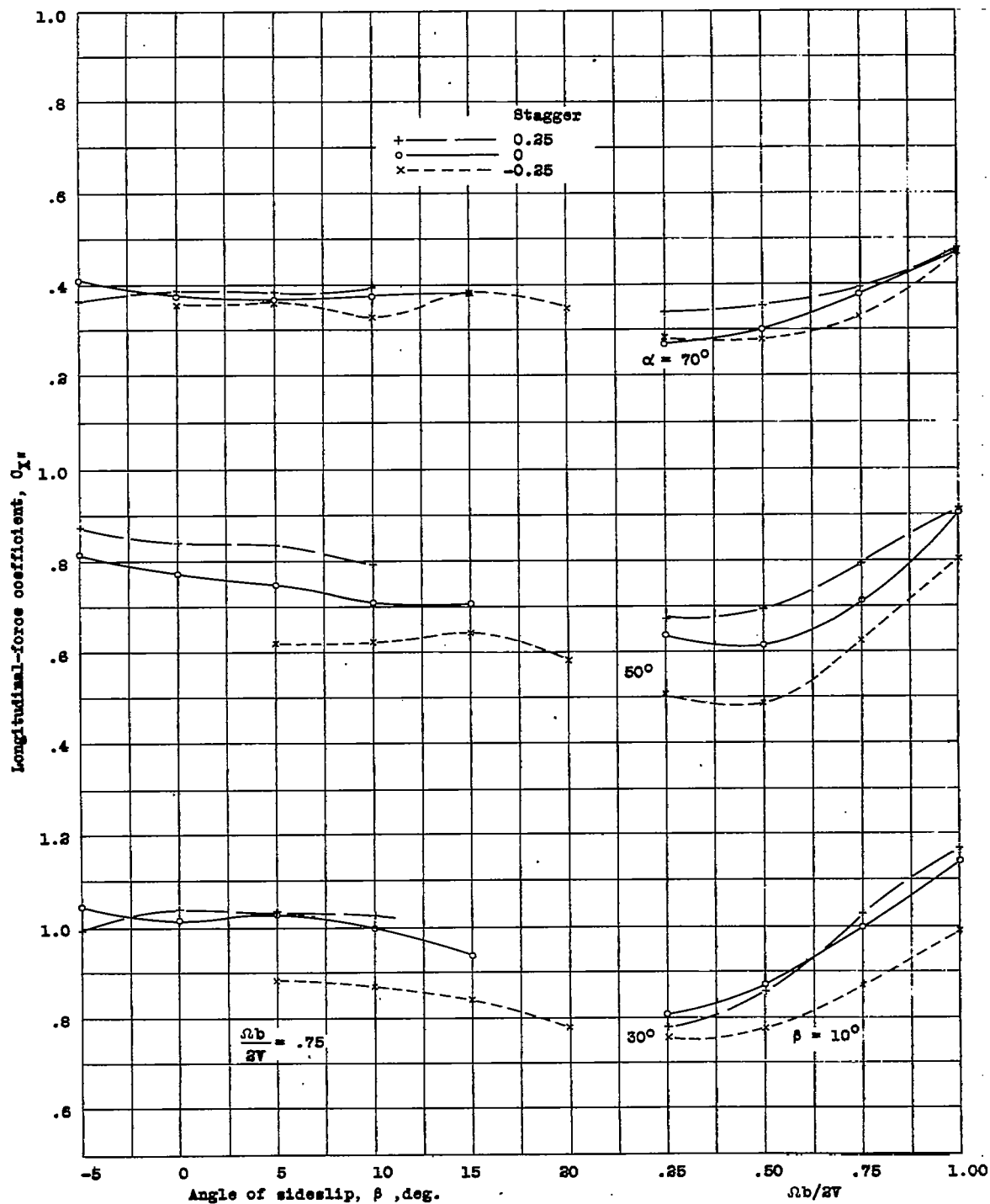mounted on the spinning balance.

Fig. 2

Figure 3.- Variation of longitudinal-force coefficient, C_X" (earth axes) with angle of sideslip and with $\Omega b/2V$.
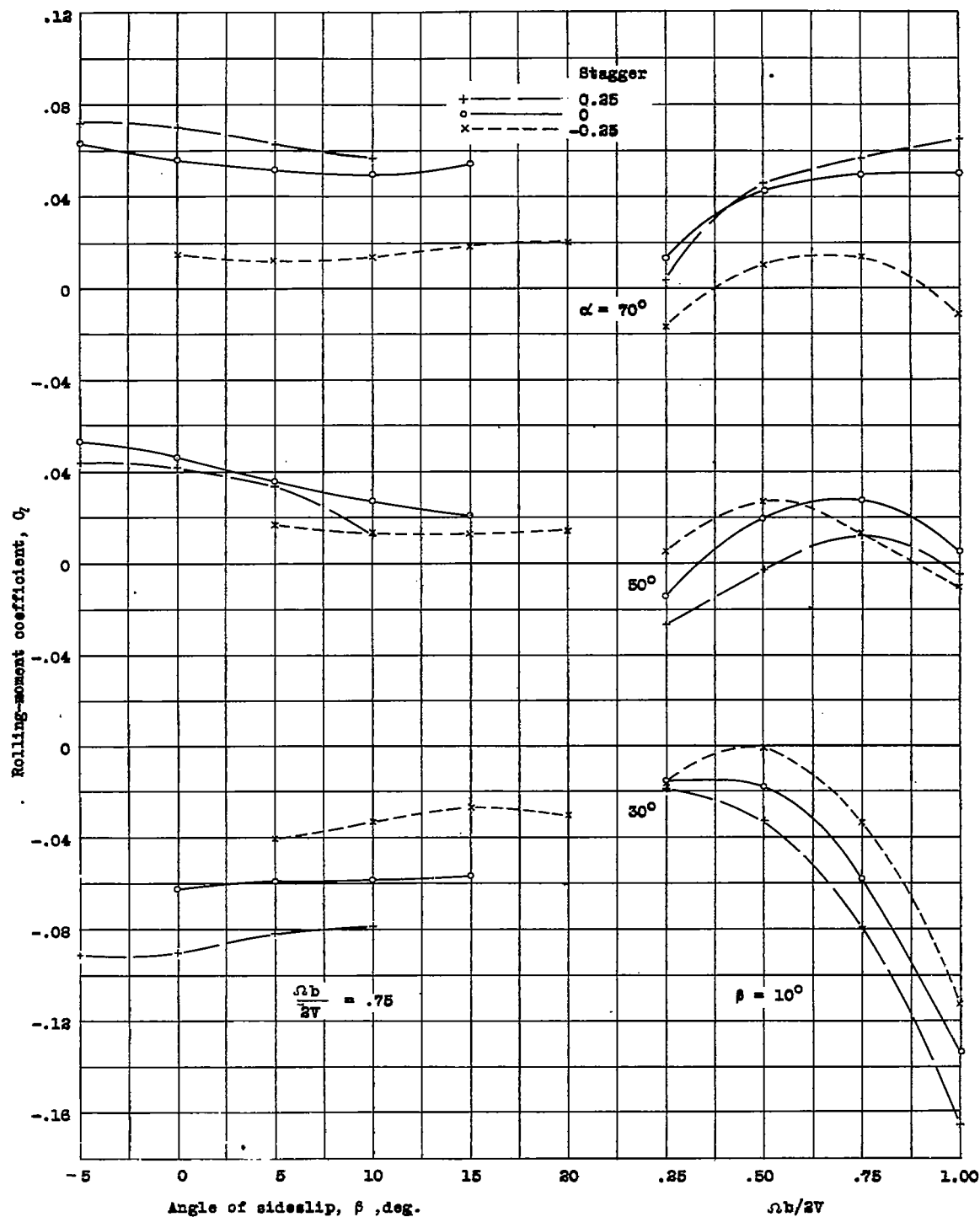
Figure 4.- Variation of rolling-moment coefficient C_l (body axes) with angle of sideslip and $\Omega b/2V$.
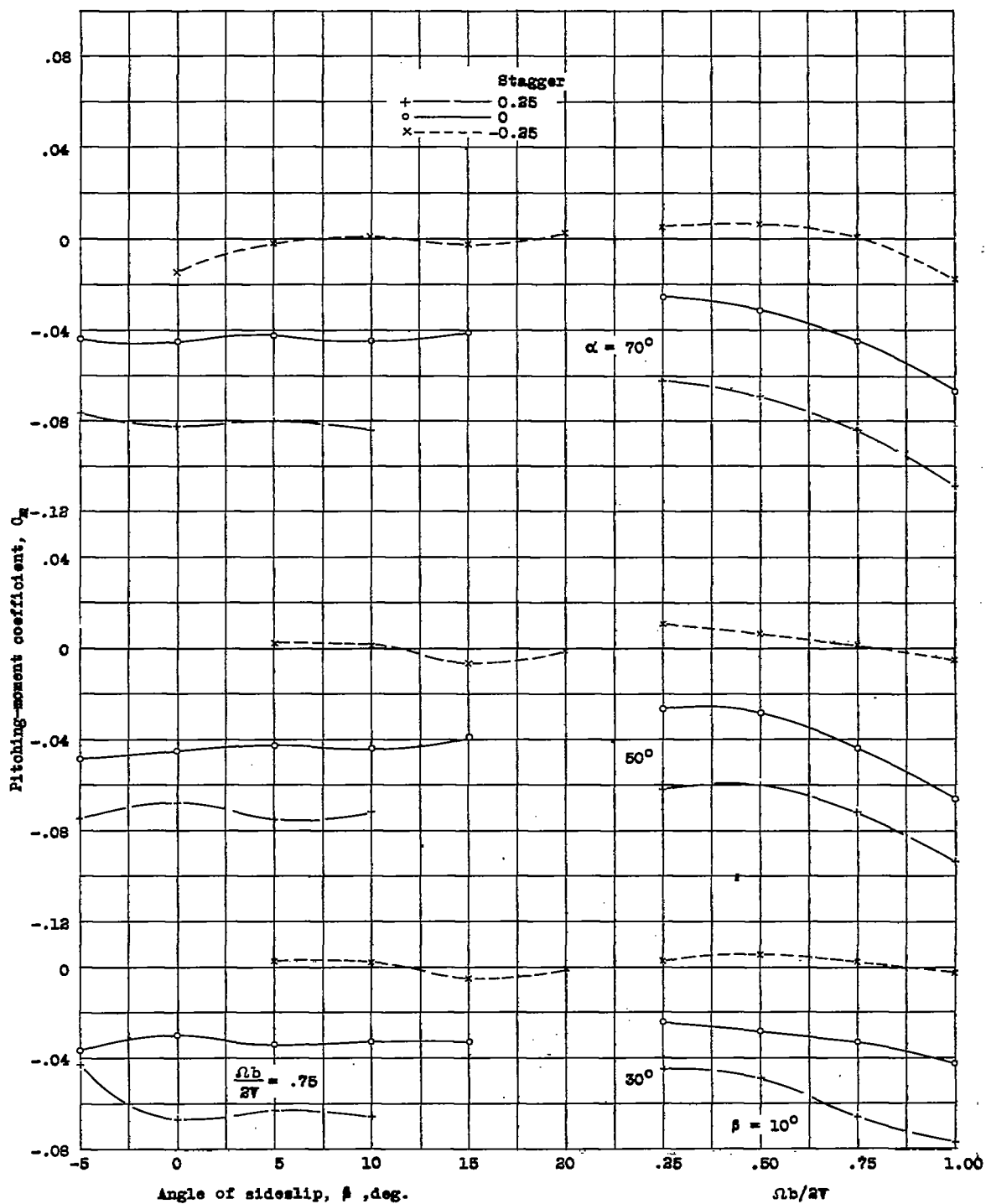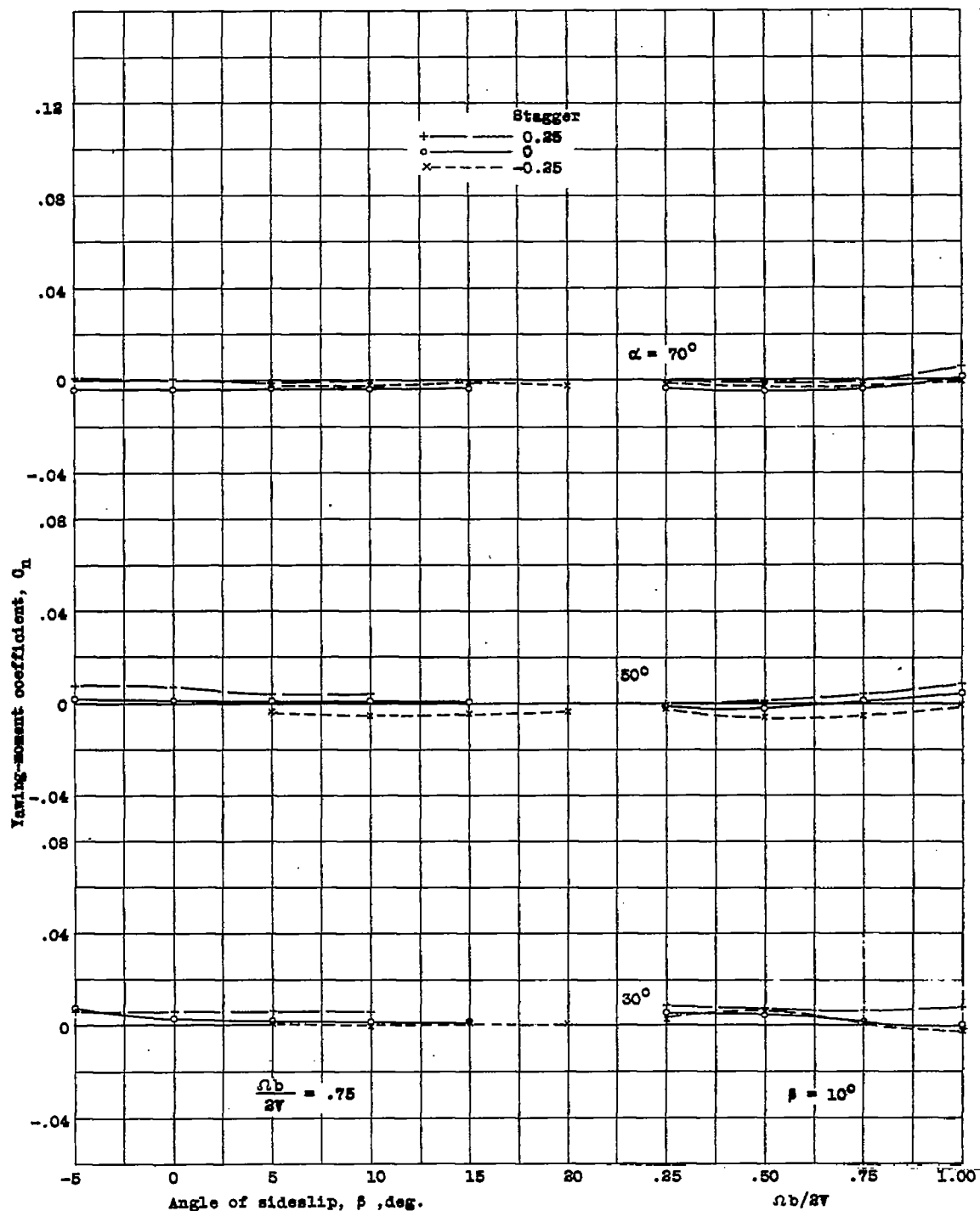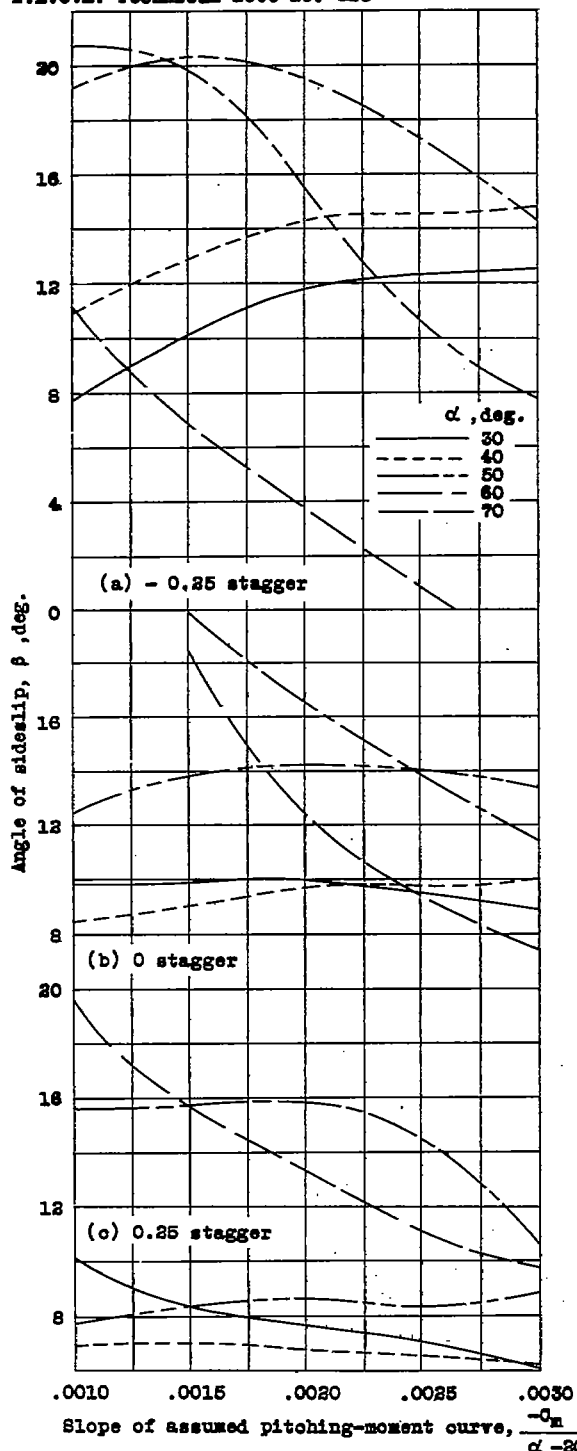
Figure 5.- Variation of pitching-moment coefficient C_m (body axes) with angle of sideslip and $\Omega b/2V$.

Figure 6.- Variation of yawing-moment coefficient C_n(body axes) with angle of sideslip and with $\Omega b/2V$.

Figure 7.- Effect of pitching-moment coefficient upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_{X}$

$$\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0 \qquad \qquad \frac{b^2}{k_Z^2 - k_X^2} = 80$$
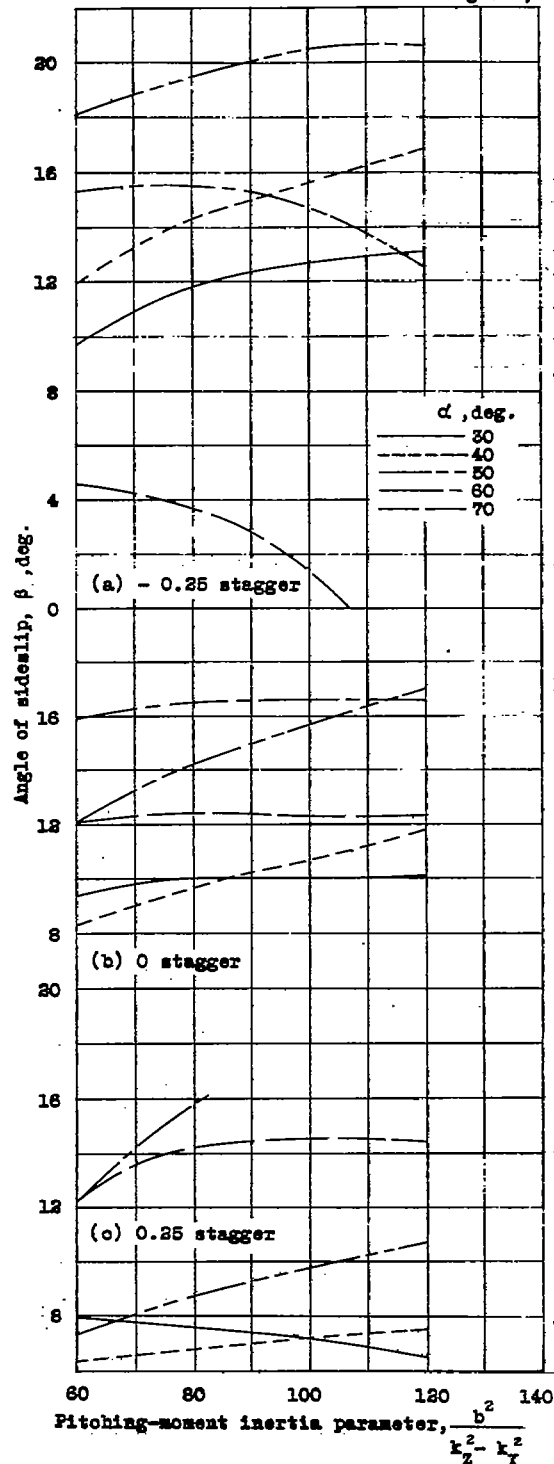
Figure 8.- Effect of pitching-moment inertia parameter upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_{X}$

$$C_m = -0.0020(\alpha - 20^\circ), \frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$$
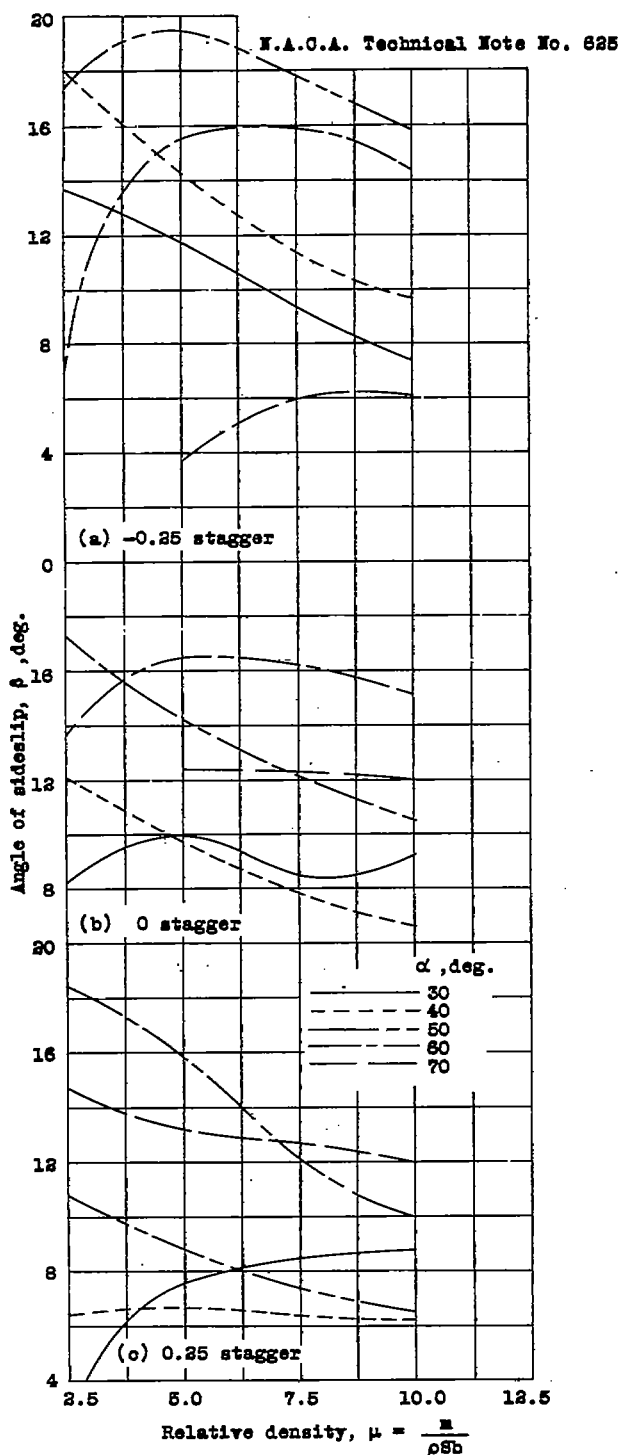
Figure 9.- Effect of relative density of airplane upon sideslip necessary for equilibrium in a spin. $C_m = -0.0020(\alpha - 20^\circ)$

$$C_L = C_{X^*} \quad \frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0 \qquad \frac{b^2}{k_Z^2 - k_X^2} = 80$$

Rolling-moment and yawing-moment inertia

$$\text{parameter,} \quad \frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$$
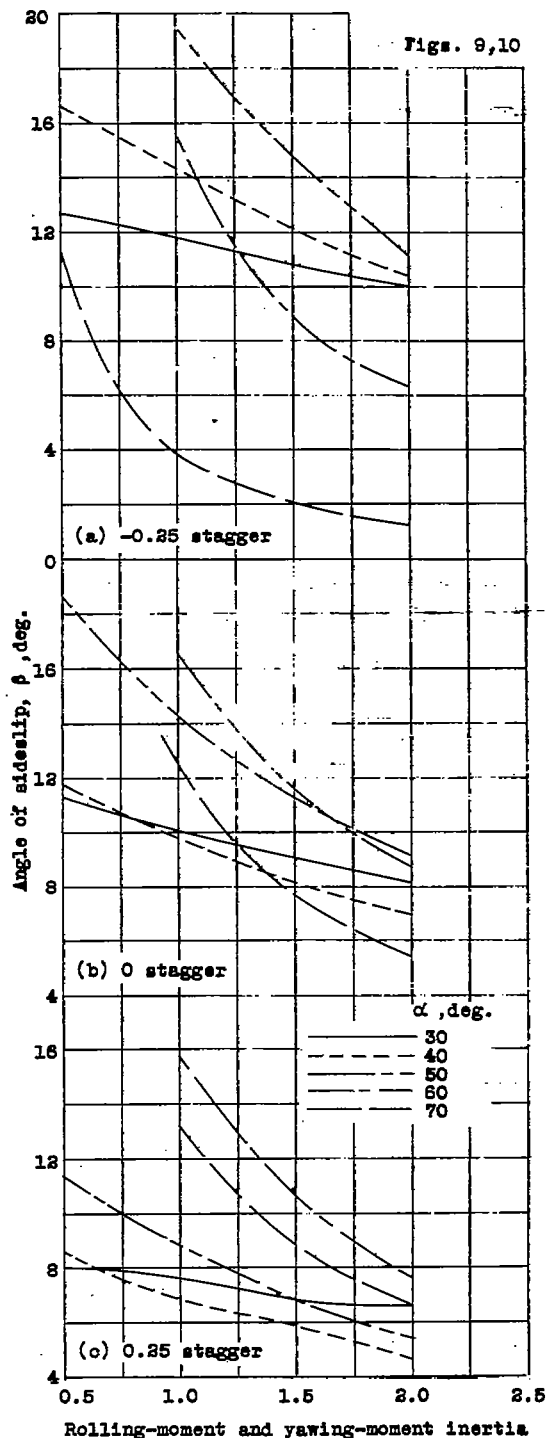
Figure 10.- Effect of rolling-moment and yawing-moment inertia parameter upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_{X^*}$ $\frac{b^2}{k_Z^2 - k_X^2} = 80$

$$C_m = -0.0020(\alpha - 20^\circ)$$
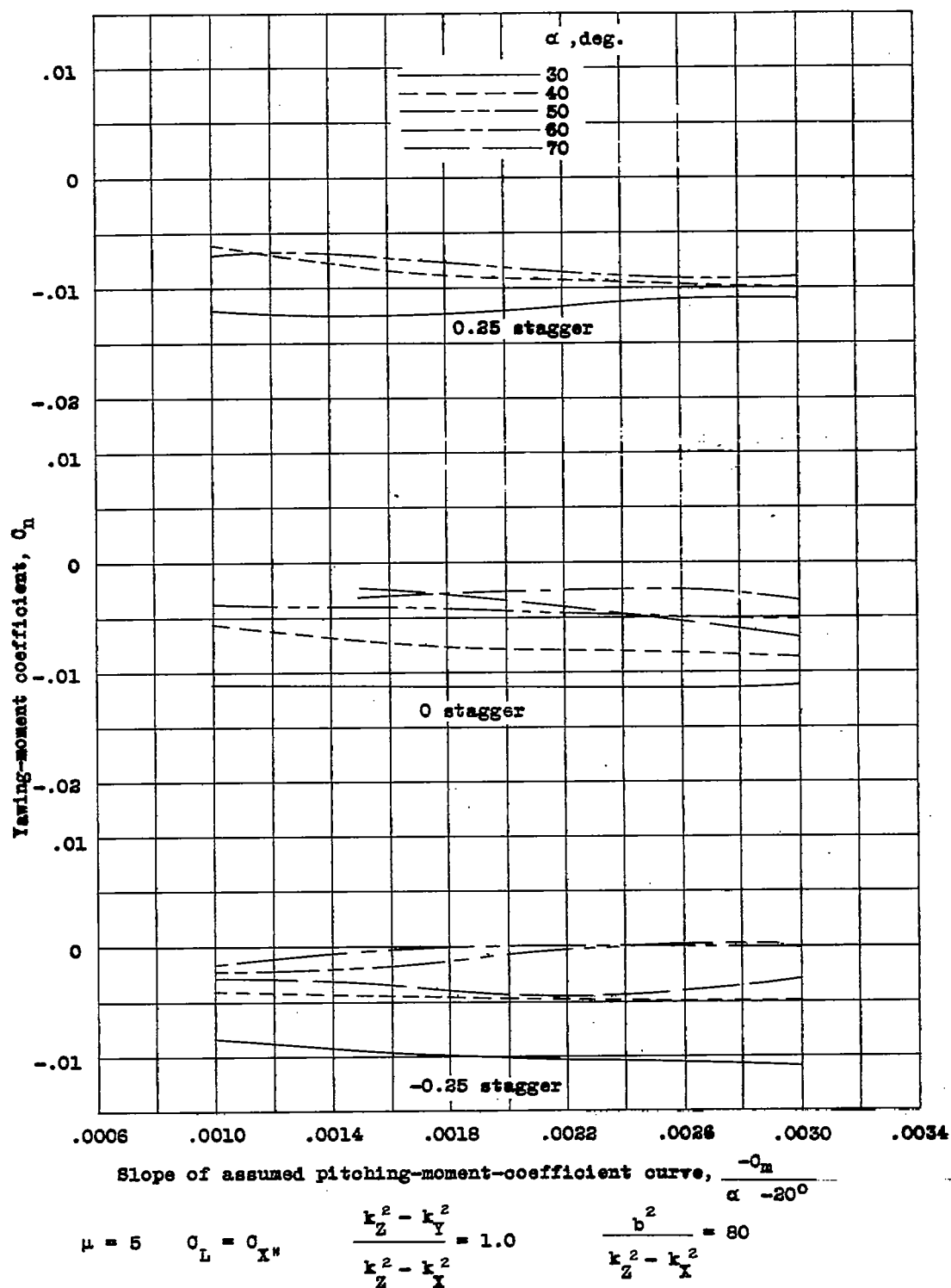
Figure 11.- Effect of pitching-moment coefficient upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin.
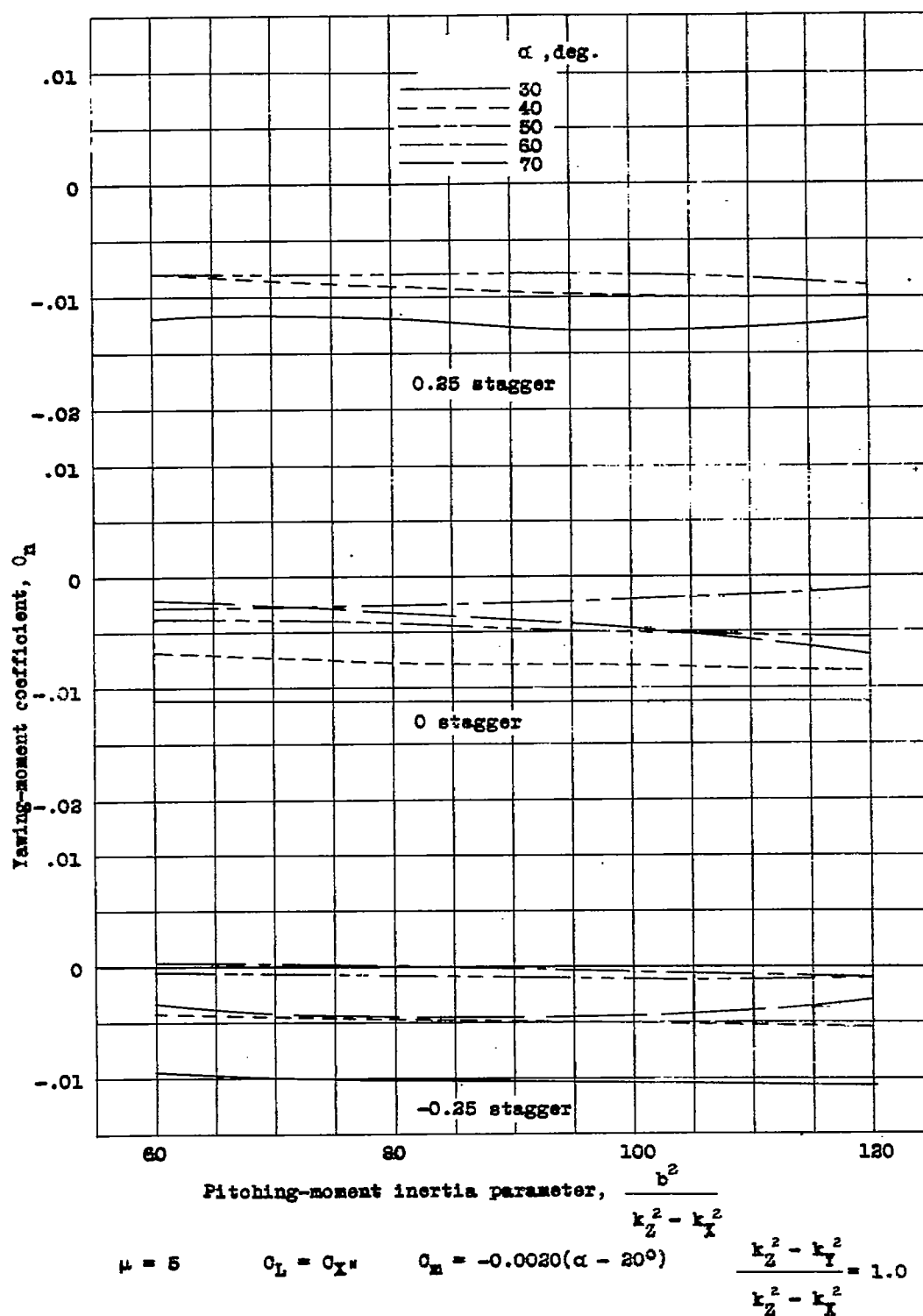
Figure 12.- Effect of pitching-moment inertia parameter upon yawing-moment
coefficient that must be supplied by parts other than the wing
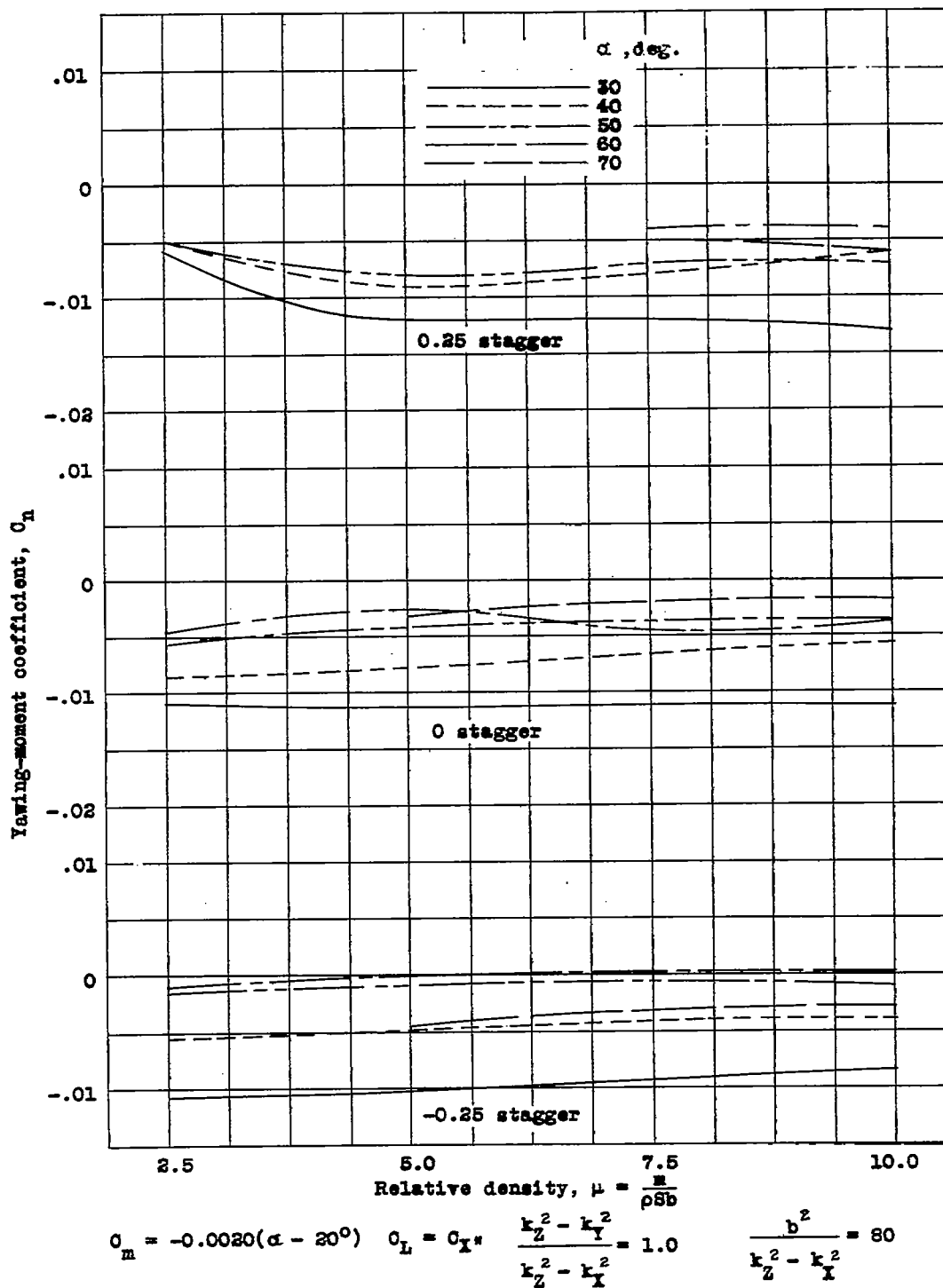for equilibrium in a spin.

Figure 13.- Effect of relative density of airplane upon yawing-moment
         coefficient that must be supplied by parts other than the
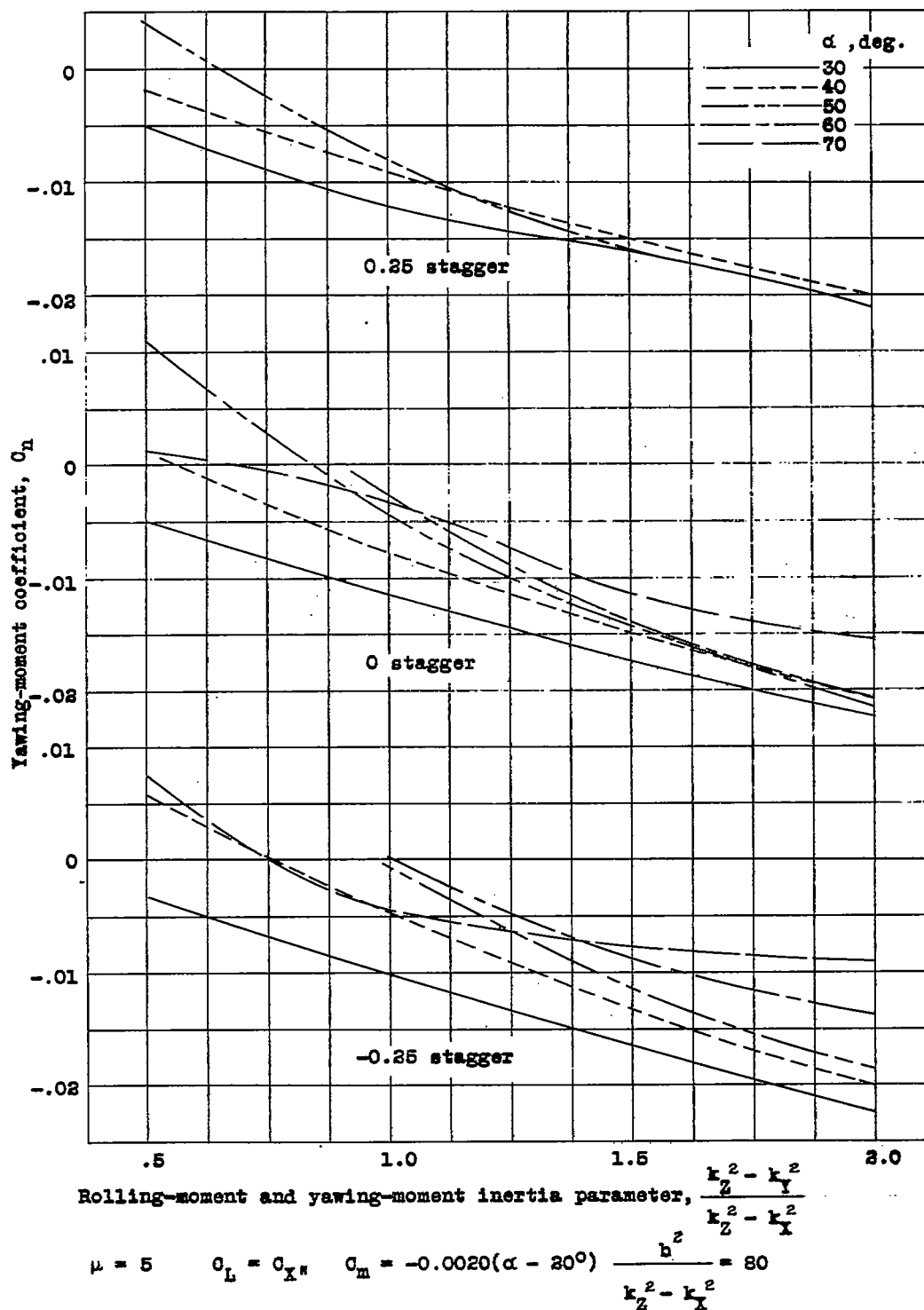wing for equilibrium in a spin.

Figure 14.- Effect of rolling-moment and yawing-moment inertia parameter
upon yawing-moment coefficient that must be supplied by parts
other than the wing for equilibrium in a spin.